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RESEARCH MEMORANDUM

A PRELIMINARY FLIGHT INVESTIGATION OF THE EFFECTS OF VORTEX GENERATORS ON SEPARATION DUE TO SHOCK

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NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

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A PRELIMINARY FLIGHT INVESTIGATION OF THE EFFECTS OF VORTEX GENERATORS ON SEPARATION DUE TO SHOCK

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SUMMARY

A preliminary investigation was made of an airplane in flight to determine the effectiveness of vortex generators in preventing separation due to compressibility shock at full-scale Reynolds numbers. Several arrangements of vortex generators were tested at 10 and 30 percent chord on the upper surface between 44 and 61 percent semispan on a wing of an F-51D airplane. The wing contour over this portion of the span was modified for other tests and indicated earlier flow separation than the original contour. The vortex generators, consisting of small airfoils of rectangular plan form mounted normal to the surface, were arranged in a spanwise row and set at an angle of attack of about 15°, so as to produce a system of vortices for the purpose of increasing the transfer of momentum into the turbulent boundary layer. The effectiveness of the generators was determined at supercritical speeds (Mach numbers 0.71 to 0.77) and over a range of lift coefficients from measurements of chordwise pressure distributions and from boundary-layer measurements made at the trailing edge.

The results of the tests indicated that all the arrangements of vortex generators tested delayed separation to higher Mach numbers or lift coefficients. The vortex generators mounted at both 10 and 30 percent chord however were considerably more effective than vortex generators at 10 percent chord and delayed separation at a Mach number of 0.745 beyond a lift coefficient of 0.53, and the lift coefficient at a given value of integrated total-head loss was increased as much as 0.35 above that for the basic configuration. A single row of vortex generators at 30 percent chord was about as effective as vortex generators mounted at both 10 and 30 percent chord for the range of flight conditions at which comparisons could be made. With adjacent vortex generators arranged to produce vortices rotating in opposite directions, the results were somewhat more favorable than with the vortex generators arranged to produce vortices in the same direction.

More extensive investigations are required for the determination of the optimum arrangement of vortex generators.

INTRODUCTION

In the investigations of references 1 to 5, vortex generators were found to be effective in reducing or eliminating separation of turbulent boundary layers on wings and in diffusers at subsonic speeds. The vortex generators used in these investigations consisted of small airfoils mounted normal to the surface ahead of the region of separation. An application of vortex generators to flow at transonic speeds in reference 6 indicated that separation due to compressibility shock could also be reduced or eliminated. The implication of these results, particularly for airplanes at transonic speeds, is obvious since separation affects maximum lift, stability, and trim. Since the problem of buffeting is recognized as being associated with intermittent separation and reattachment of the boundary layer, it seems possible that buffeting may be reduced by the use of vortex generators. Inasmuch as the Reynolds number of the tests of reference 6 was low, a preliminary flight investigation reported herein was undertaken in order to determine whether vortex generators were also effective at full-scale Reynolds number. The tests were made on an F-51D airplane at critical speeds and over a range of lift coefficients. The tests included several arrangements of vortex generators mounted on the upper surface of one wing and over only a portion of the span. The effectiveness of the generators was determined from pressure distributions over part of the wing chord and boundary-layer surveys at the trailing edge.

SYMBOLS

C_L	airplane lift coefficient
c	wing chord (74.5 in. at test station)
M	Mach number
Δp_t	total-pressure loss
q_c	free-stream impact pressure
x	distance along chord from leading edge
y	distance above surface

Subscripts:

o	free stream
δ	edge of boundary layer

APPARATUS AND TESTS

The vortex generators used in this investigation were small airfoils of rectangular plan form having chords of 1/2 inch and 1 inch and spans of 1/4 inch, 1/2 inch, and 3/4 inch. The generators were made of strips of steel and filed to airfoil sections approximating Clark Y and biconvex sections. Several arrangements of vortex generators were tested. The generators were spaced 2 inches center to center in a spanwise direction and extended from 44 to 61 percent semispan at 10 percent, 30 percent, and at both 10 and 30 percent chord on the upper surface of the left wing of the F-51D airplane. For the 10-percent-chord location the generators, having approximate Clark-Y sections, were welded to a steel strip 1/16 inch thick which was fastened to the surface with flush screws. The steel strip was faired into the surface over a chordwise distance of about 8 inches. At the 30-percent-chord location the generators, having approximate biconvex sections, were welded to flush-mounted flat-head screws. The generators were set at an angle of attack of 15° to produce vortices rotating in the same direction (co-rotating) or so that adjacent generators produced vortices rotating in opposite directions (counterrotating). An arrangement of generators at 10 percent chord is shown in figure 1. Details of all the arrangements of vortex generators are given in table I and sketched in figure 2.

The portion of the wing span over which the vortex generators were tested had a modified airfoil section described in reference 7 as contour B. Since the mounting plate for the generators at 10 percent chord was faired into the surface, this fairing or bump was retained for tests of the basic section without the generators.

Static-pressure measurements were made along the upper surface of the wing with flush orifices between 17 and 68 percent chord. The static pressure at the trailing edge was obtained with a tube located about 1/8 inch above the wing surface with the static holes about 1/8 inch forward of the trailing edge. Total pressure through a portion of the boundary layer of the upper surface was measured by a rake of total-pressure tubes extending about 3 inches above the surface and located at the trailing edge of the wing at the center line of the arrangement of vortex generators. The rake is shown mounted on the trailing edge in figures 1 and 3. The free-stream total and static pressures were measured by means of a pitot-static head mounted on a wing boom. The position error of this installation had previously been determined.

The tests were made in dives from an altitude of 28,000 feet to about 20,000 feet. Pressures were continuously recorded from a Mach number of about 0.71 to about 0.77 in the dive and through the pull-out.

RESULTS AND DISCUSSION

Some results of the chordwise-pressure-distribution measurements for the basic configuration and the configurations with counterrotating vortex generators at 10 and 30 percent chord are presented in figures 4 to 6 in terms of chordwise variation of Mach number outside of the boundary layer, M_δ, for several free-stream Mach numbers and airplane lift coefficients. These were the only chordwise pressure distributions that were evaluated. The values of M_δ were computed using free-stream total pressure and static pressure at the surface. Results of the boundary-layer measurements at the trailing edge are given in figures 7 to 10 as distribution of Mach number through the lower portion of the boundary layer and in figures 11 to 13 as distribution of the ratio of total-pressure loss to free-stream impact pressure, $\Delta p_t/q_c$, through the boundary layer for an airplane lift coefficient of 0.17 and several Mach numbers and for a constant Mach number of 0.745 and several values of airplane lift coefficient.

The chordwise distribution of M_δ for the basic configuration (fig. 4) did not appreciably differ from that of contour B in reference 7, indicating that the fairing added at 10 percent chord had no apparent effect on the flow behind this station. As the free-stream Mach number and airplane lift coefficient were increased to values from 0.736 to 0.75 and from 0.11 to 0.17, respectively, the position of compressibility shock rapidly moved forward about 10 percent of the chord and separated flow occurred behind the shock, as is indicated by the higher Mach numbers near the trailing edge and also by the boundary-layer characteristics in figures 7 and 11. With a single row of counterrotating vortex generators at 10 percent chord, the chordwise distribution of M_δ (fig. 5) was similar to that for the basic configuration except that the forward movement of the shock and flow separation were delayed to higher airplane lift coefficients (also see figs. 8 and 12). Since the other arrangements of vortex generators at 10 percent chord (Nos. 2 and 3) had similar boundary-layer characteristics, they are not presented. For counterrotating generators at 30 percent of the chord (No. 5), the chordwise distribution of M_δ was considerably modified. In addition to the main shock at about 54 percent chord, several smaller shocks are apparent in figure 6, associated perhaps with local flow disturbance due to the vortex generators. The position of the main shock appears to be practically fixed at about 54 percent chord for the flight conditions of the tests. The boundary-layer characteristics in figures 9 and 13 show that there is no separation up to a Mach number of 0.745 and a lift coefficient of 0.35. This was the highest C_L obtained at $M_o = 0.745$ for this configuration; however, a lift coefficient of 0.54 at $M_o = 0.745$ was obtained when testing vortex generators located at both 10 and 30 percent chord (No. 4). A comparison between configurations 4 and 5 cannot be drawn at $C_L = 0.54$, but it is interesting to note (fig. 10) that the distribution of Mach number through the boundary layer for configuration 4 gives no indication of separation. The boundary-layer characteristics for the co-rotating configuration at 30 percent chord (No. 6) are similar to those for configurations 4 and 5 and are therefore not presented.

The relative effectiveness of all of the arrangements of vortex generators is indicated qualitatively in figures 14 to 16 by comparisons of the ratio of total-pressure loss to free-stream impact pressure integrated over the height of the survey rake. The rake was not large enough to survey completely the wake; therefore, the integrated areas can only be used as a qualitative indication of total-head loss in the wake. A greater percentage of the total-head loss was measured in the case of a small loss than in the case of a large total-head loss. The integrated area tended to approach a maximum value and remain constant when the flow was separated. Results are plotted for constant lift coefficient and varying Mach number and for constant Mach number and varying lift coefficient in each figure. The test points for which the boundary layer was definitely separated at the trailing edge are indicated by tails. It is believed that separation may have started at lower values of Mach number and lift coefficient than is indicated in the figures.

The effects of vortex generator size and/or spacing at 10 percent chord are illustrated in figure 14. The double row of vortex generators (No. 3) at 10 percent chord resulted in the greatest reduction of total-pressure loss. The single row of the largest vortex generators (No. 2) showed some tendency toward a greater reduction of total-pressure loss at lift coefficients above about 0.30, but the tests of that configuration did not cover a range of lift coefficients sufficient to make the data conclusive.

A change of chordwise location of the vortex generators from 10 to 30 percent chord (fig. 15) showed a greater reduction of total-pressure loss than that produced by changes in size or spacing at 10 percent chord. A single row of vortex generators at 30 percent chord was about as effective as vortex generators mounted at both 10 and 30 percent chord for the range of flight conditions at which comparisons could be made. The results in figure 15 show that with the generators at 30 percent chord, separation due to shock was delayed beyond a lift coefficient of 0.35 at a Mach number of 0.745 and the lift coefficient at a given value of integrated total-head loss was increased about 0.2 above that for the basic configuration. Vortex generators at both 10 and 30 percent chord delayed separation due to shock beyond a lift coefficient of 0.53 at $M_o = 0.745$ and the lift coefficient at a given value of integrated total-head loss was increased as much as 0.35 above that for the basic configuration.

The comparisons in figure 15 show that the counterrotating arrangement of vortex generators at 30 percent chord was slightly more effective than the co-rotating arrangement.

CONCLUDING REMARKS

The results of the investigation indicated that all the arrangements of vortex generators tested delayed separation to higher Mach numbers or lift coefficients. The vortex generators mounted at both 10 and 30 percent chord, however, were considerably more effective than vortex generators at 10 percent chord and delayed separation at a Mach number of 0.745 beyond a lift coefficient of 0.53, and the lift coefficient at a given value of integrated total-head loss was increased as much as 0.35 above that for the basic configuration. A single row of vortex generators at 30 percent chord was about as effective as vortex generators mounted at both 10 and 30 percent chord for the range of flight conditions at which comparisons could be made. With adjacent vortex generators arranged to produce vortices rotating in opposite directions, the results were somewhat more favorable than with the vortex generators arranged to produce vortices rotating in the same direction.

More extensive investigations will be required for the determination of the optimum arrangement of vortex generators.

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1. Taylor, Harlan D.: The Elimination of Diffuser Separation by Vortex Generators. Rep. R-4012-3, United Aircraft Corp. Res. Dept., June 10, 1947.

2. Taylor, H. D.: Increasing the Efficiency of the U:A.C. 8-Ft. Wind Tunnel Fan by Means of Vortex Generators. Rep. R-4012-4, United Aircraft Corp. Res. Dept., Nov. 5, 1947.

3. Taylor, H. D.: Design Criteria for and Applications of the Vortex Generator Mixing Principle. Rep. M-15038-1, United Aircraft Corp. Res. Dept., Feb. 16, 1948.

4. McCurdy, W. J.: Investigation of Boundary Layer Control of an N.A.C.A. 16-325 Airfoil by Means of Vortex Generators. Rep. M-15038-3, United Aircraft Corp. Res. Dept., Dec. 3, 1948.

5. Taylor, H. D.: Application of Vortex Generator Mixing Principle to Diffusers. Concluding Report. Air Force Contract W33-038 ac-21825. U.A.C. Rep. R-15064-5, United Aircraft Corp. Res. Dept., Dec. 31, 1948.

6. Donaldson, Coleman duP.: Investigation of a Simple Device for Preventing Separation Due to Shock and Boundary-Layer-Interaction. NACA RM L50B02a, 1950.

7. Zalovcik, John A., and Luke, Ernest P.: Some Flight Measurements of Pressure-Distribution and Boundary-Layer Characteristics in the Presence of Shock. NACA RM L8C22, 1948.

TABLE I.- VORTEX-GENERATOR CONFIGURATIONS

Configuration number	Vortex rotation	Chordwise location on airplane (percent)	Number of rows	Approximate airfoil section	Semispan (in.) (a)	Chord (in.)	Maximum thickness (in.)
1	Counter-rotating	10	1	Clark Y	0.50	0.50	0.06
2	Counter-rotating	10	1	Clark Y	.75	1.00	.12
3	Counter-rotating	10	2	Clark Y	Forward row .25 Aft row .50	.50	.06
4	Counter-rotating	10	2	Clark Y	Forward row .25 Aft row .50	.50	.06
		30	1	Biconvex	.50	.50	.06
5	Counter-rotating	30	1	Biconvex	.50	.50	.06
6	Co-rotating	30	1	Biconvex	.50	.50	.06

[a]Semispan measured from the tip to the surface of the airplane wing.

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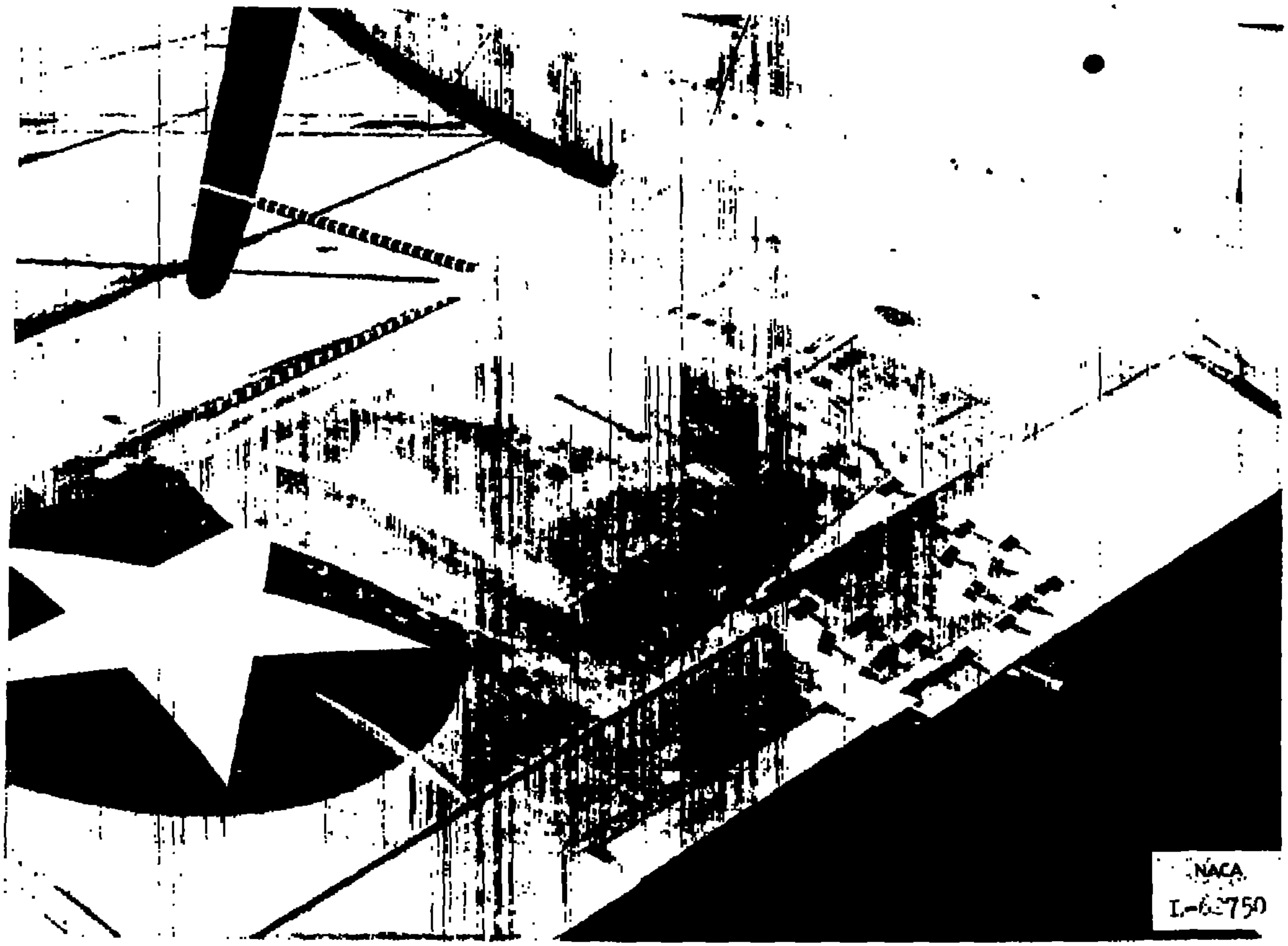


Figure 1.- F-51D airplane with pressure rake at trailing edge and vortex generators shown mounted at 10 percent chord.

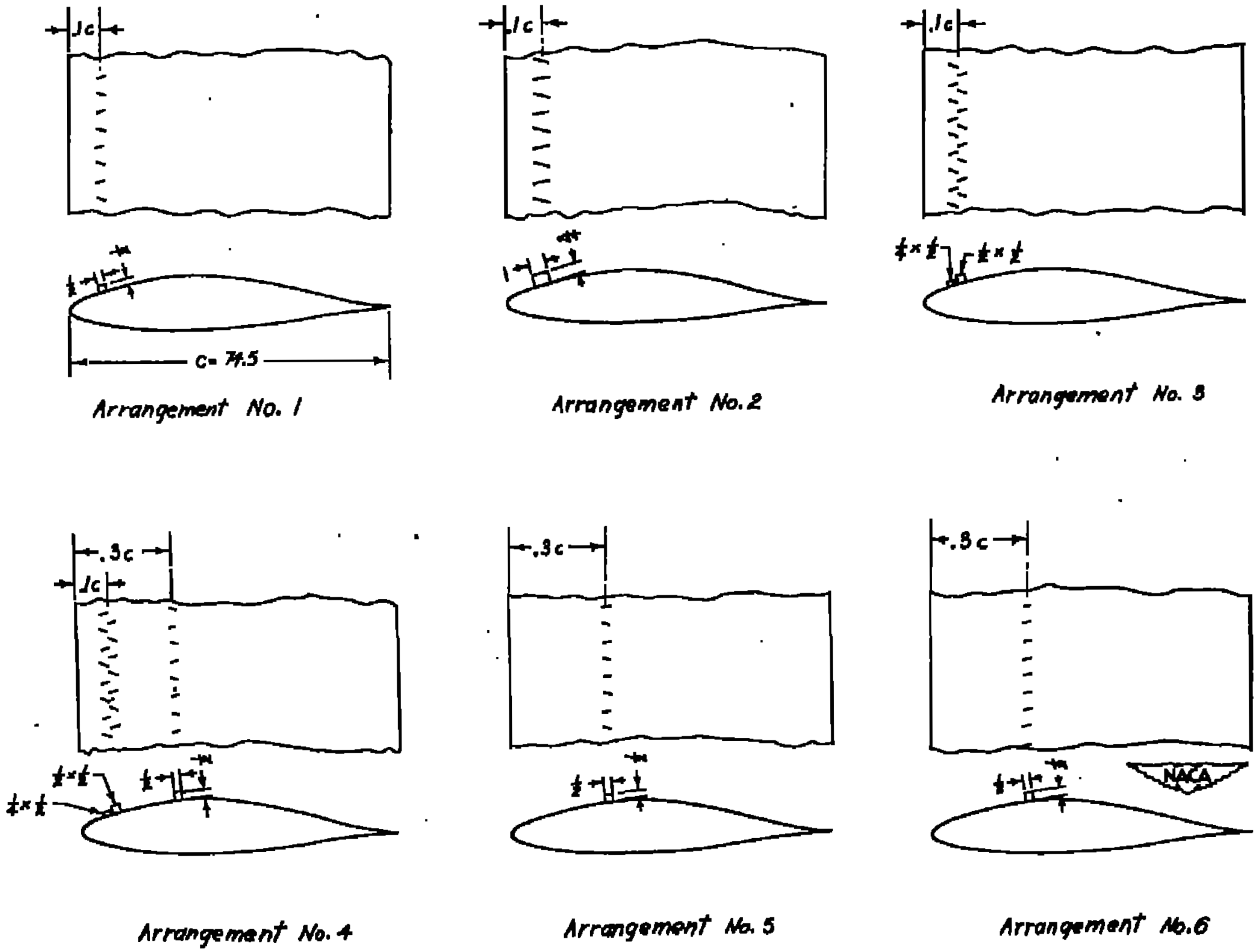


Figure 2.- Arrangements of vortex generators tested in flight. Vortex generators are drawn to a larger scale than the airplane wing. All dimensions are in inches.

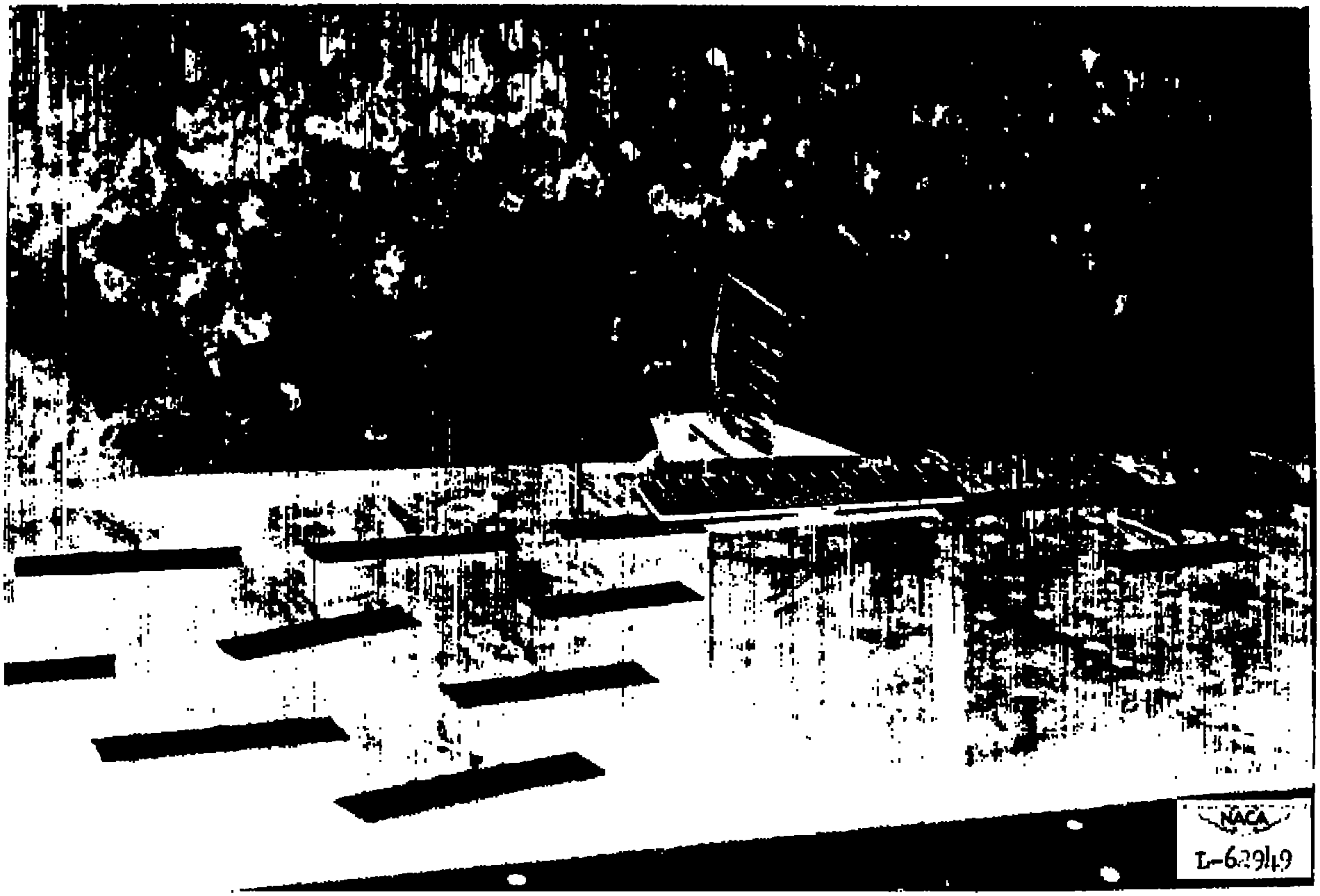


Figure 3.- Total-pressure rake and static-pressure tube at trailing edge of the wing.

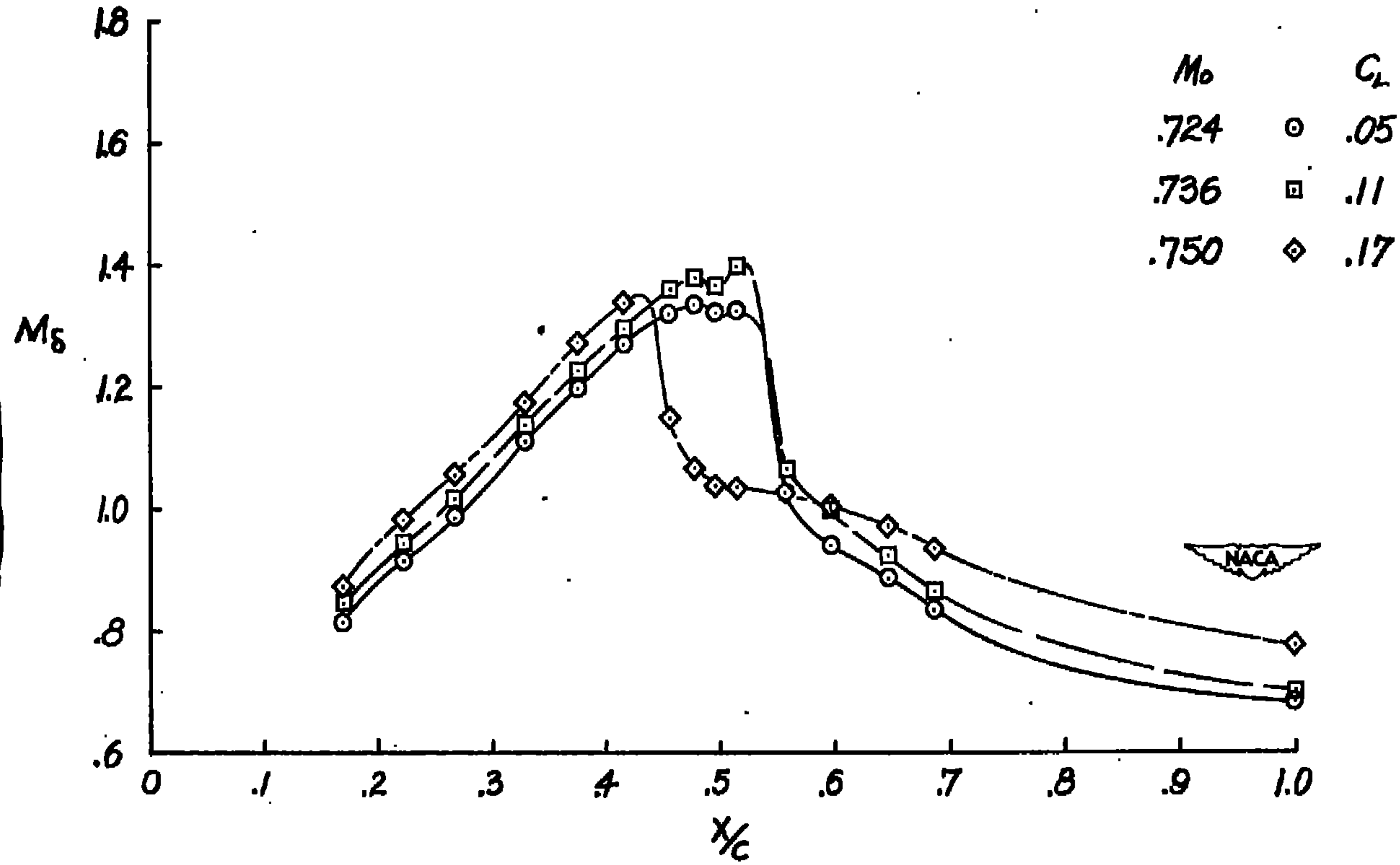


Figure 4.- Distribution of M_δ on upper surface of wing for three values of flight Mach number and airplane lift coefficient. Basic configuration.

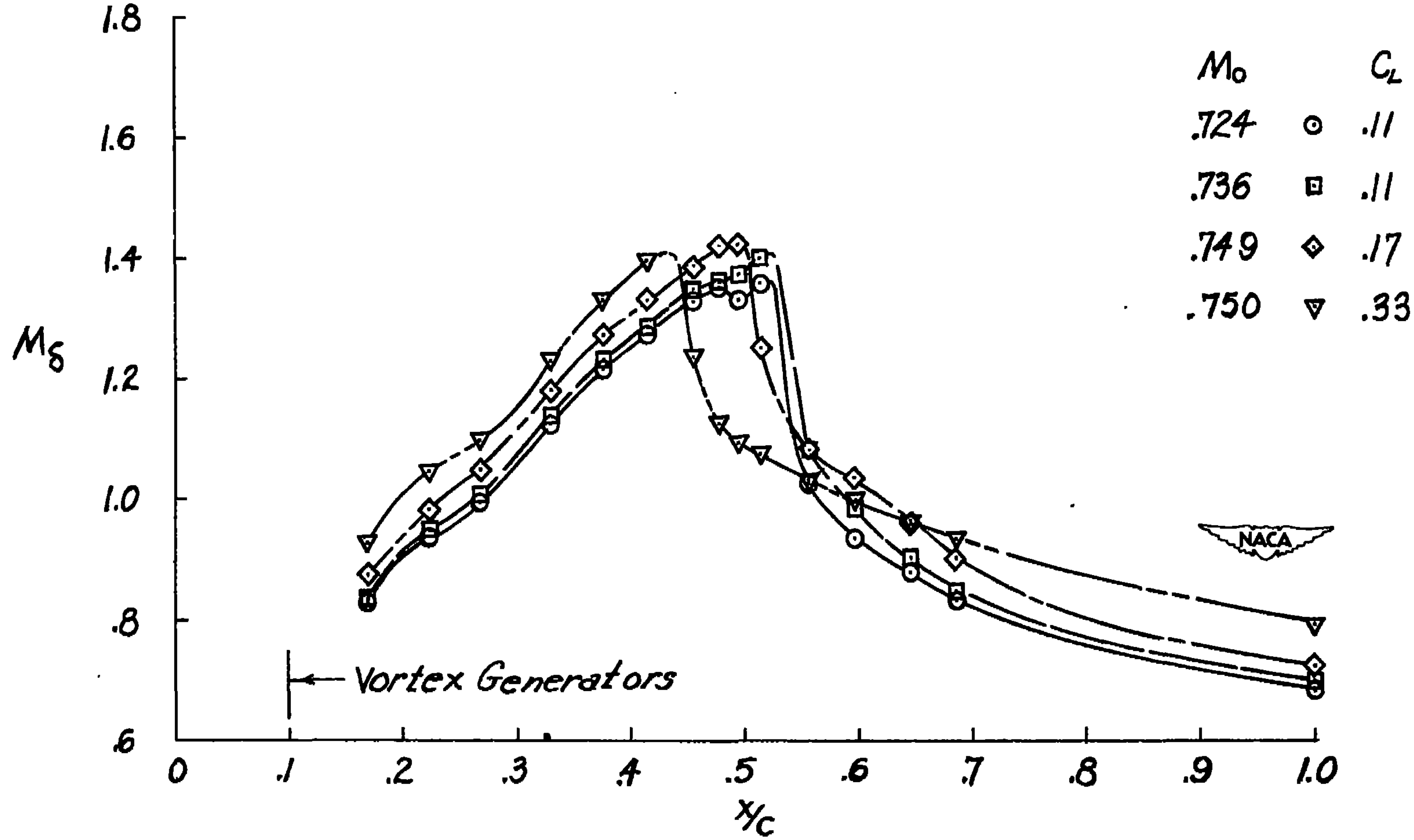


Figure 5.- Distribution of M_δ on upper surface of wing for four values of flight Mach number and airplane lift coefficient. Vortex-generator arrangement number 1.

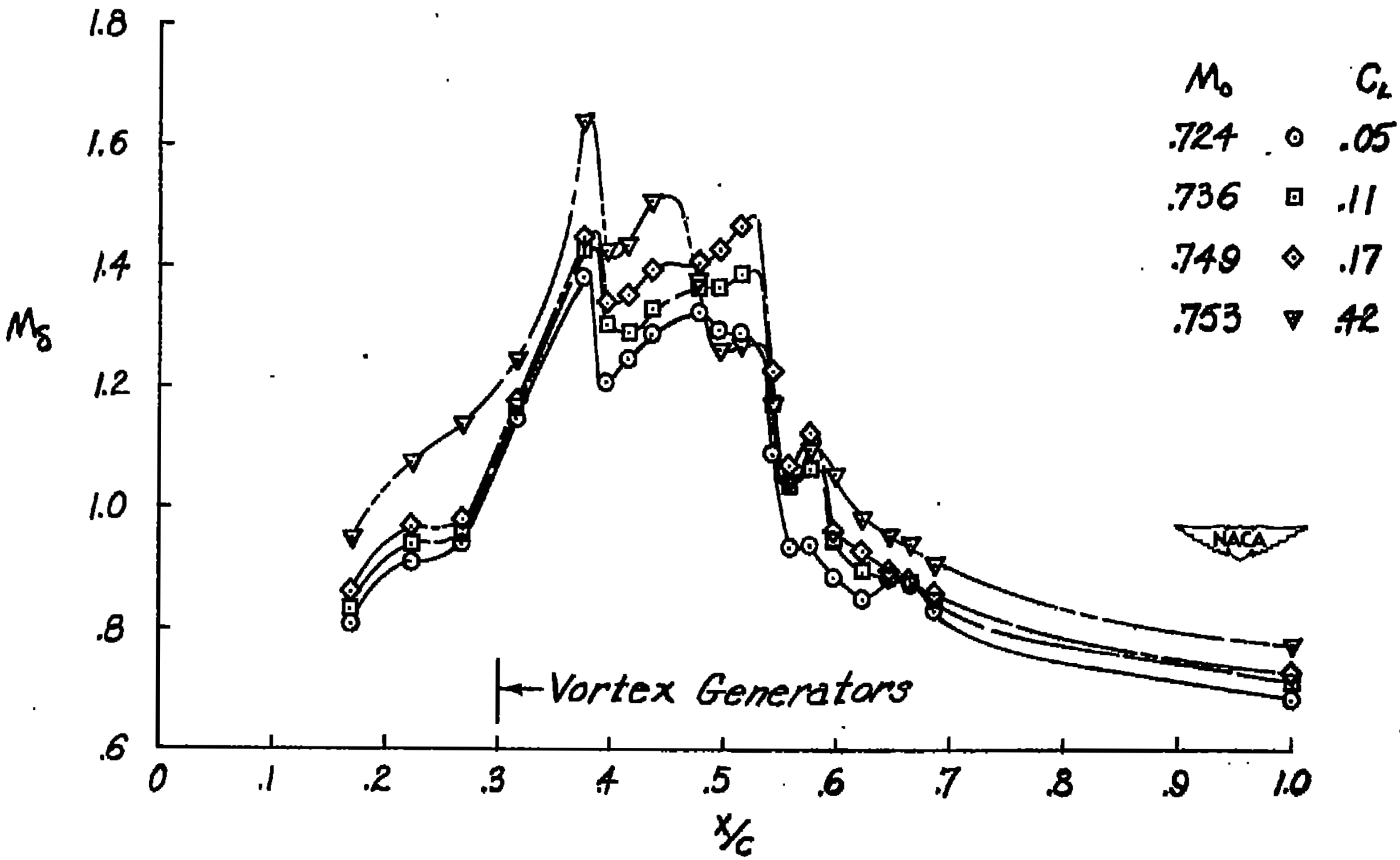
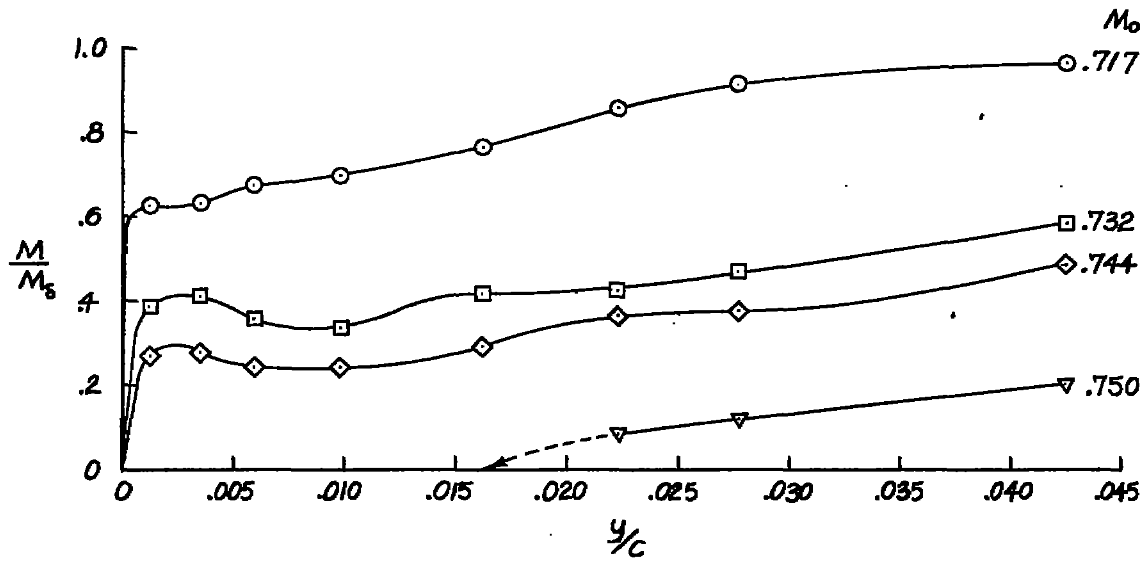
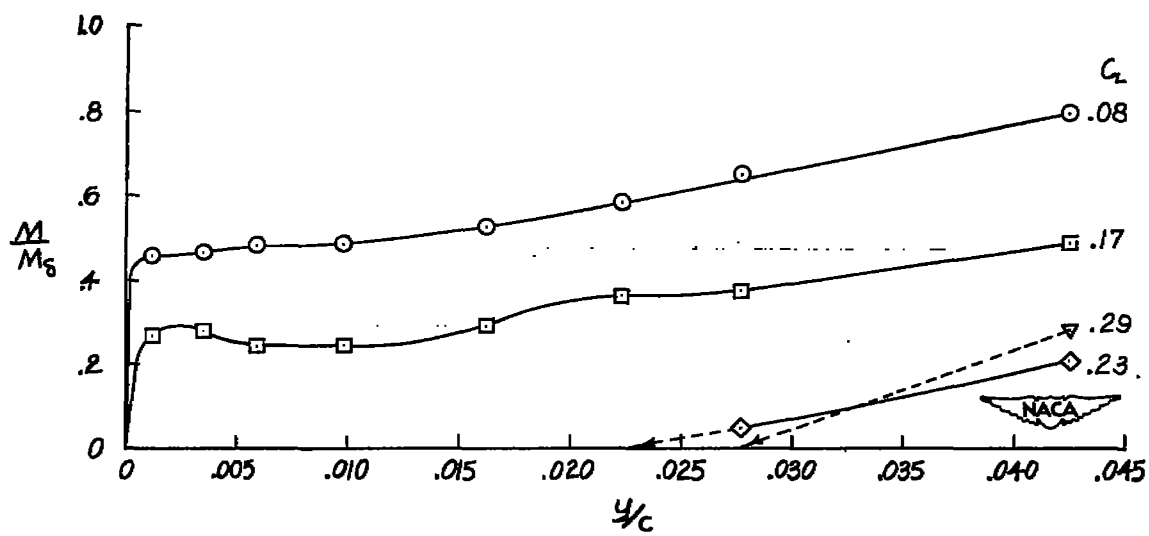


Figure 6.- Distribution of M_δ on upper surface of wing for four values of flight Mach number and airplane lift coefficient. Vortex-generator arrangement number 5.

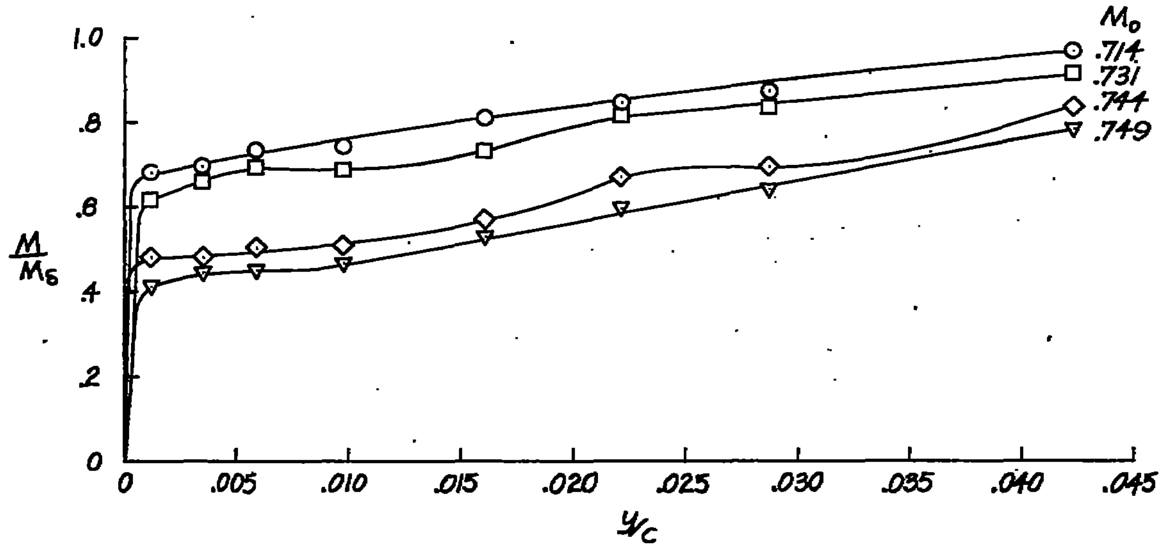


(a) $C_L = 0.17$.

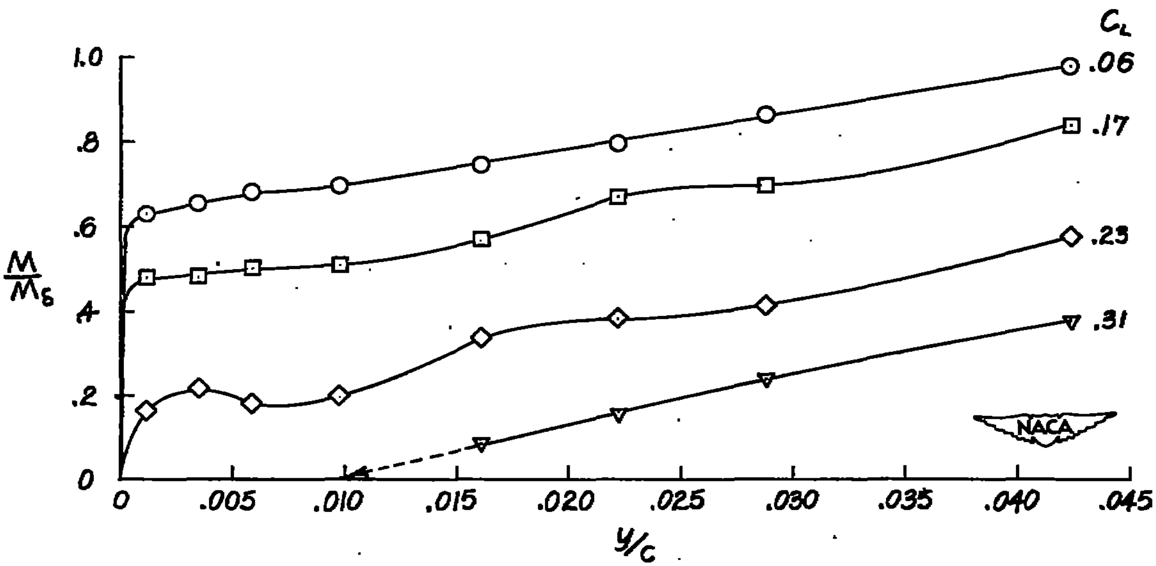


(b) $M_0 = 0.745$.

Figure 7.- Variation of Mach number through the lower portion of the boundary layer at the trailing edge. Basic configuration.

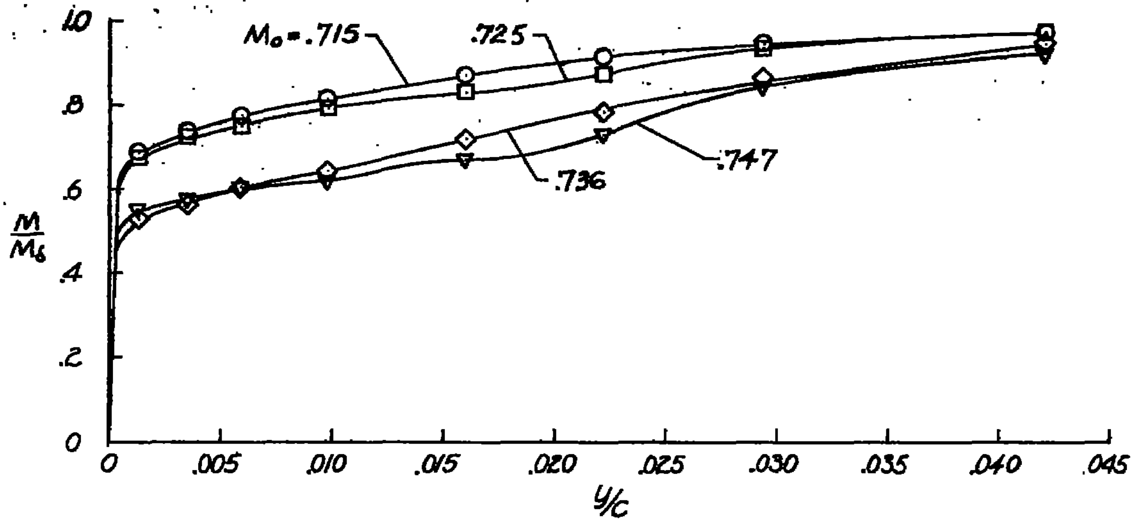


(a) $C_L = 0.17$.

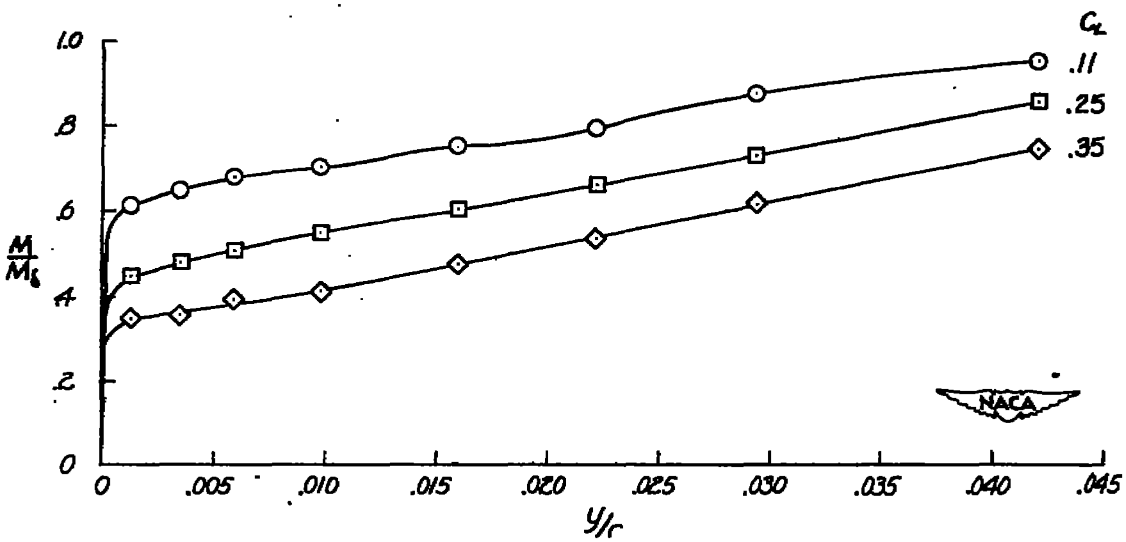


(b) $M_0 = 0.745$.

Figure 8.- Variation of Mach number through the lower portion of the boundary layer at the trailing edge. Vortex-generator configuration number 1. Single row of counterrotating generators located at 10 percent chord.

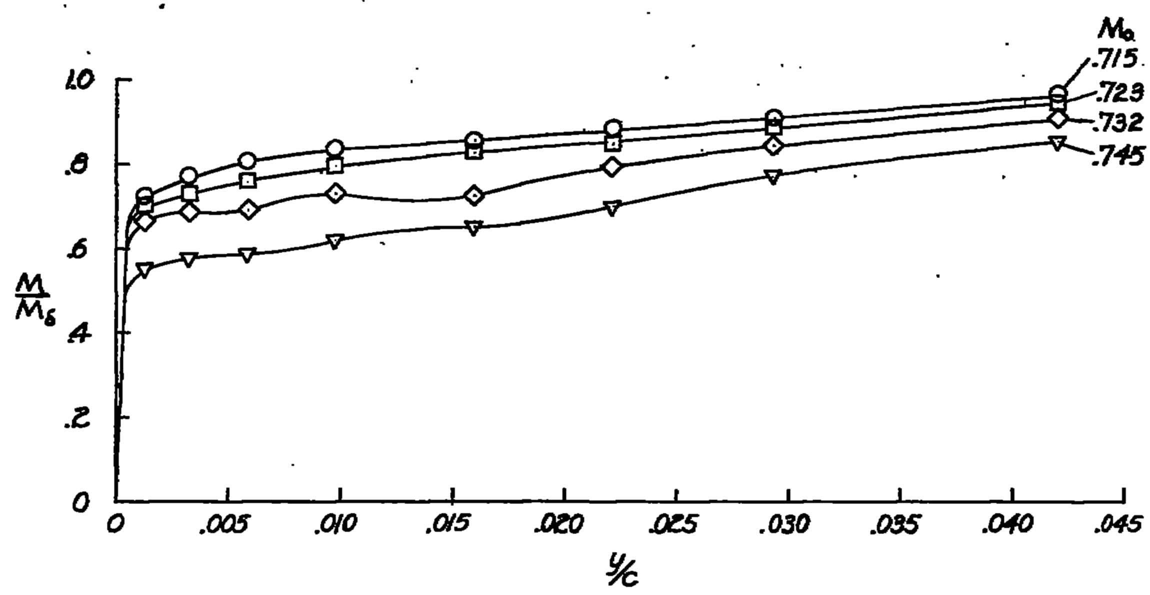


(a) $C_L = 0.17$.

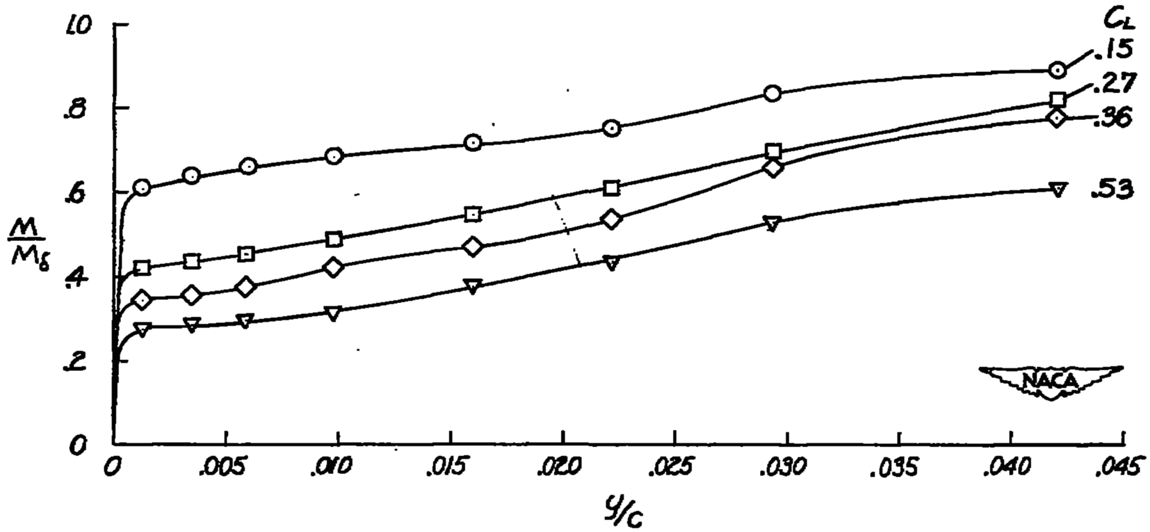


(b) $M_0 = 0.745$.

Figure 9.- Variation of Mach number through the lower portion of the boundary layer at the trailing edge. Vortex-generator configuration number 5. Single row of counterrotating vortex generators located at 30 percent chord.

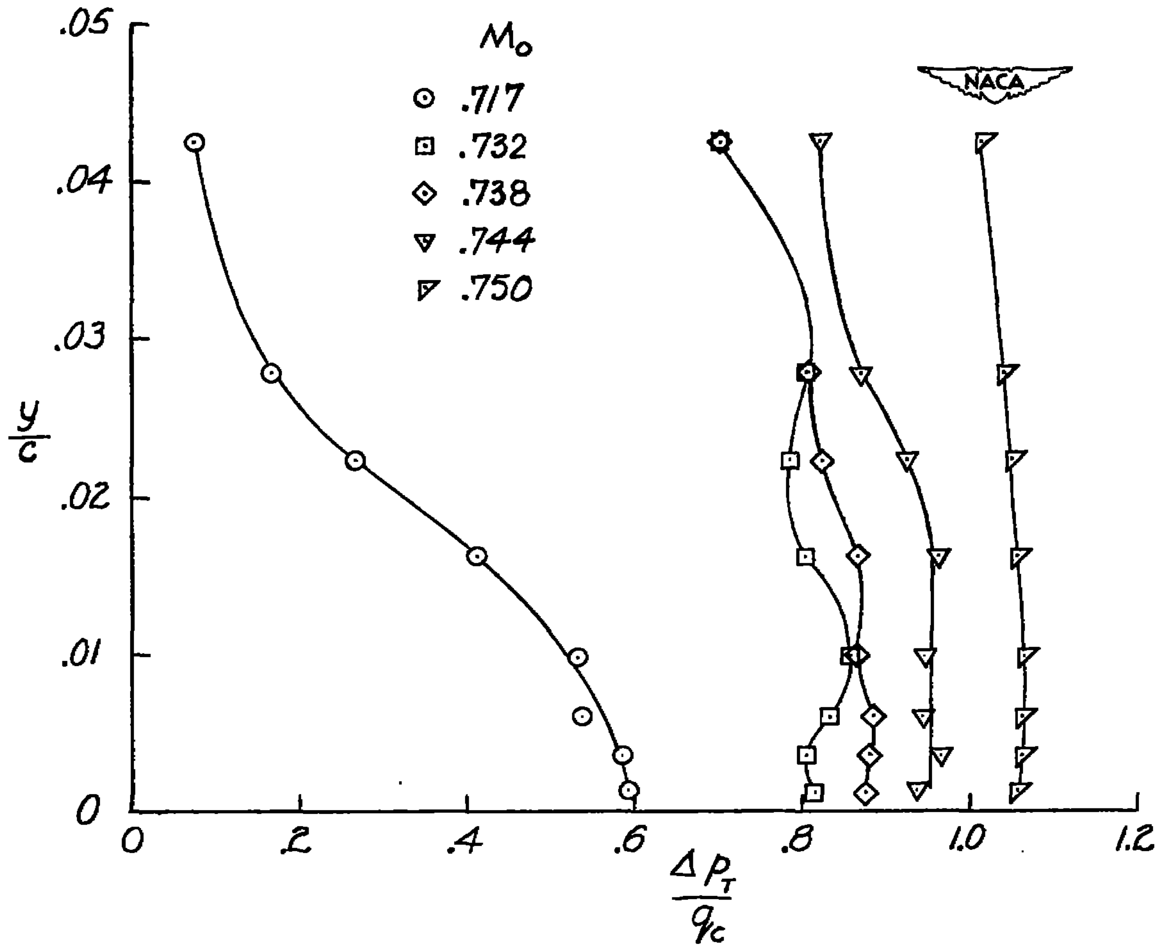


(a) $C_L = 0.17$.



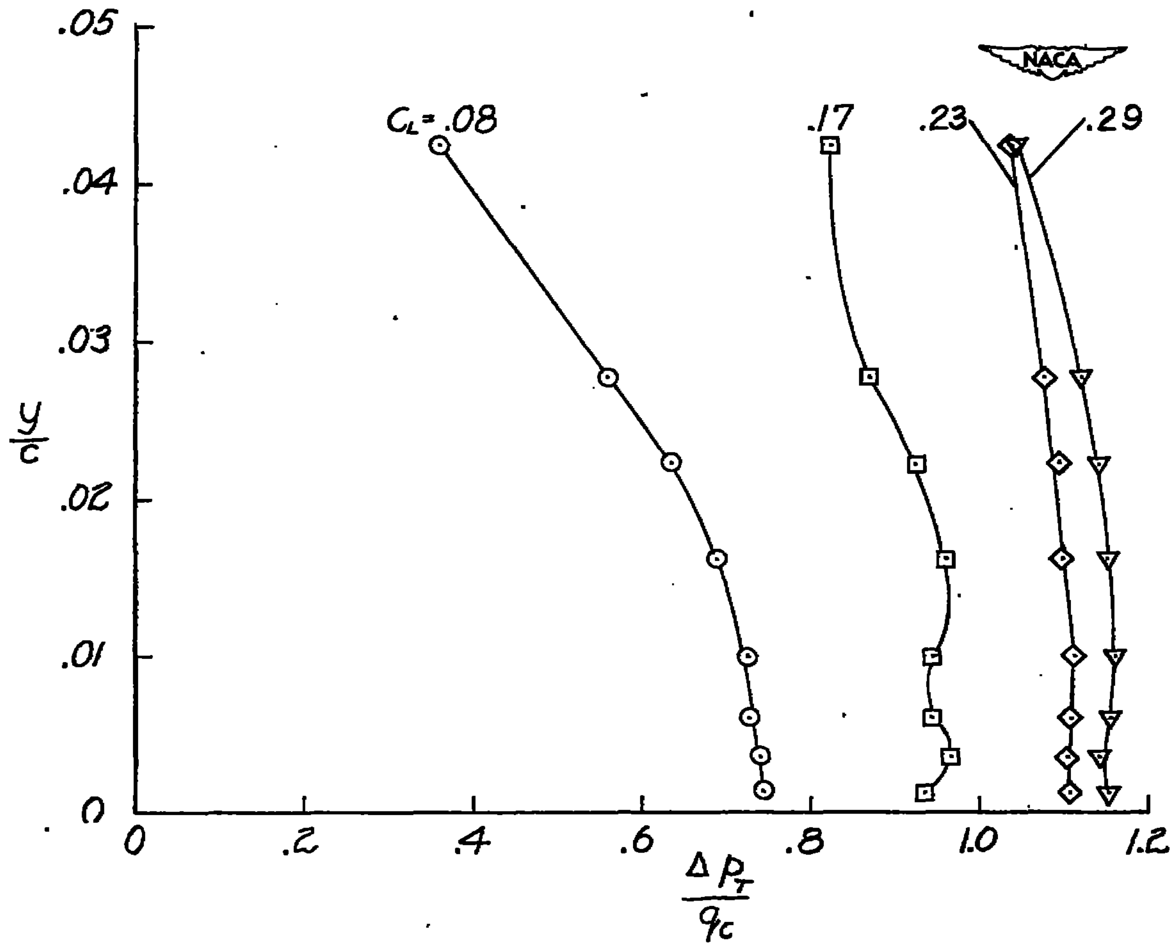
(b) $M_0 = 0.745$.

Figure 10.- Variation of Mach number through the boundary layer at the trailing edge. Vortex-generator configuration number 4. Counterrotating double row of vortex generators at 10 percent chord and counterrotating single row of vortex generators at 30 percent chord.



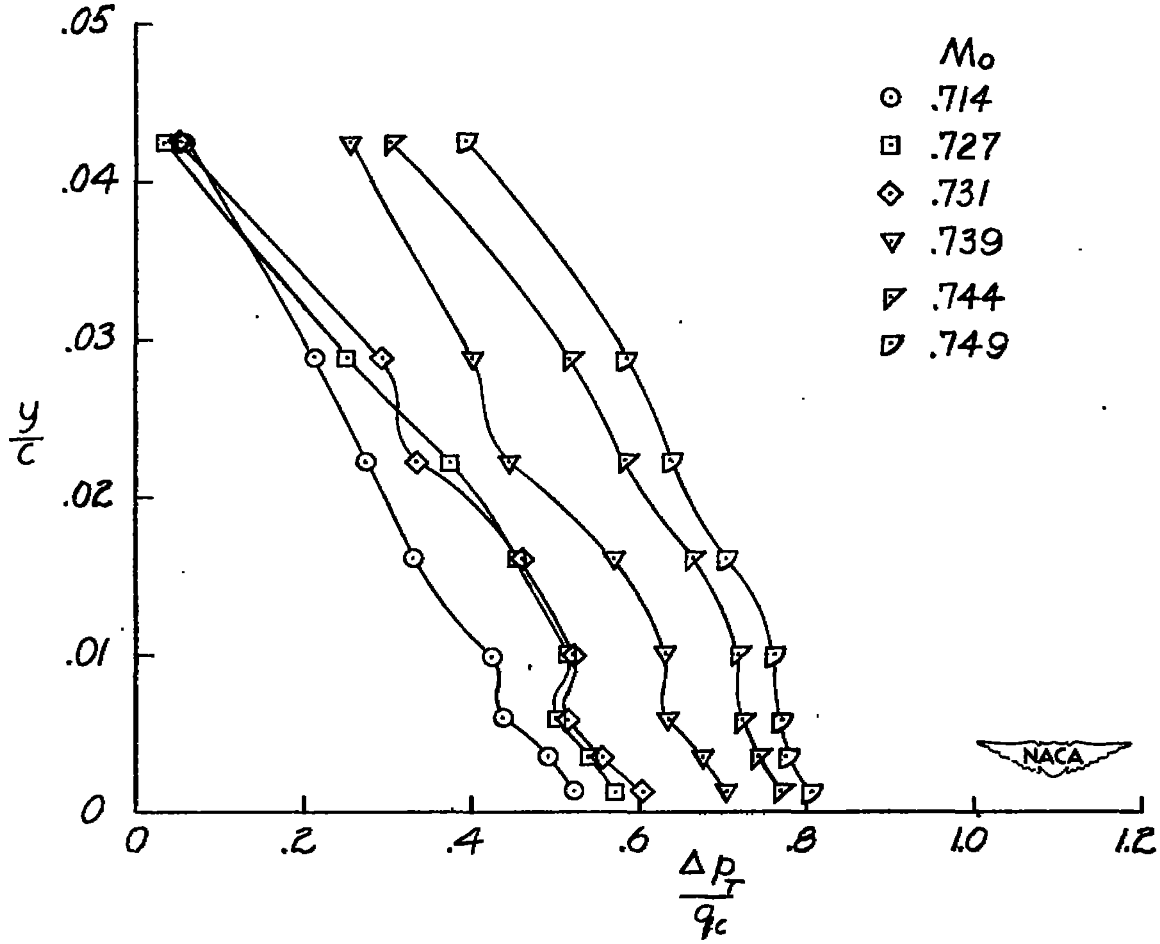
(a) $C_L = 0.17$.

Figure 11.- Variation through part of the wake at the trailing edge of the ratio of total-head loss to free-stream impact pressure. Basic configuration.



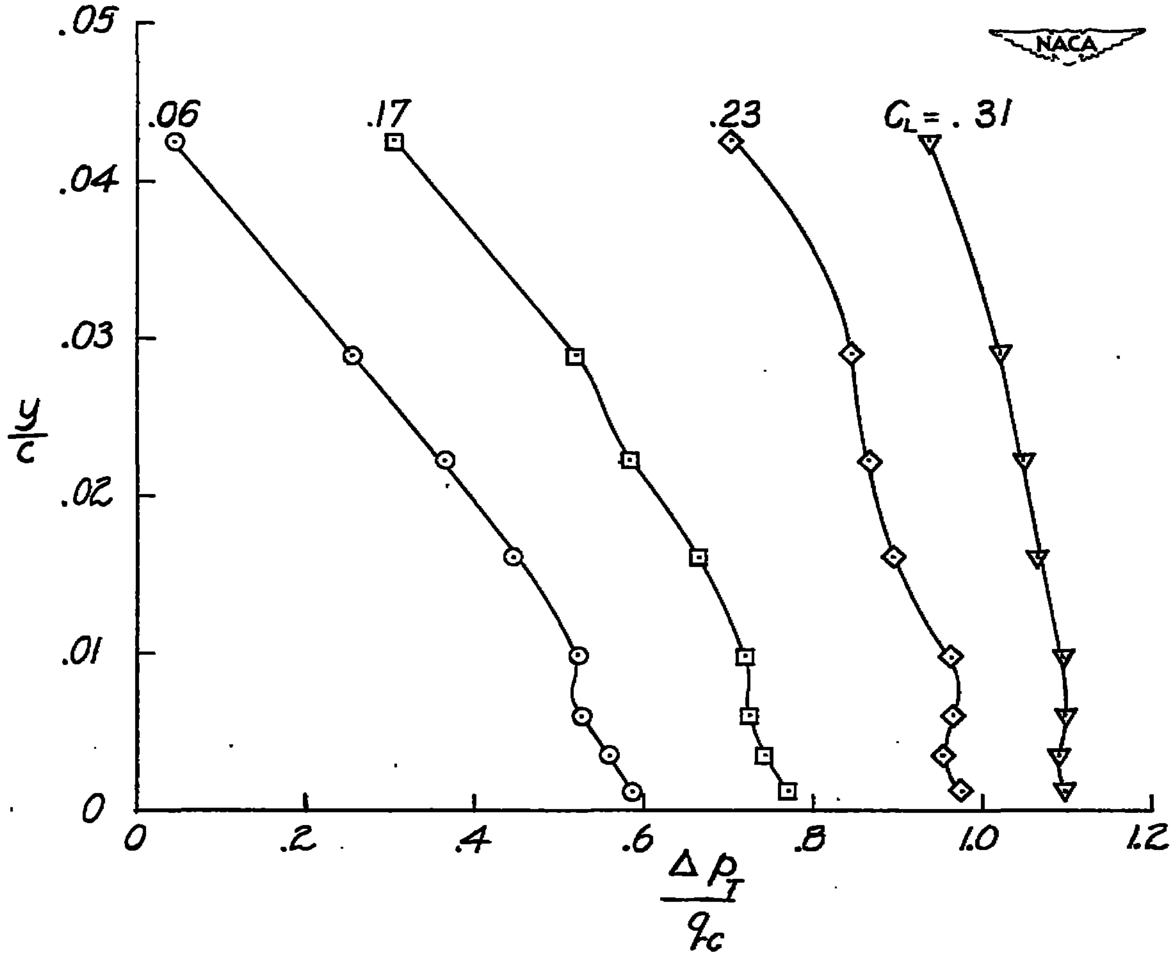
(b) $M_0 = 0.745$.

Figure 11.- Concluded.



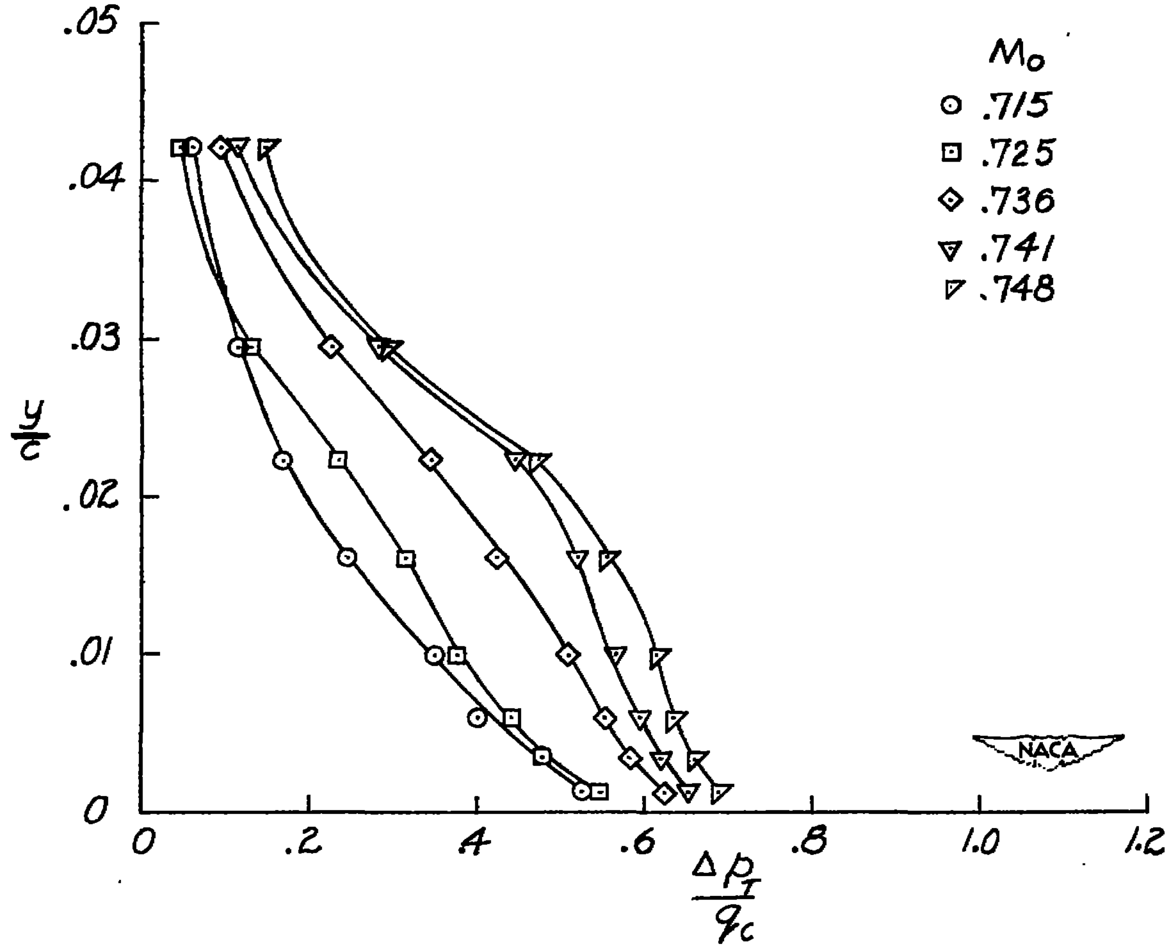
(a) $C_L = 0.17$.

Figure 12.- Variation through part of the wake at the trailing edge of the ratio of total-head loss to free-stream impact pressure. Vortex generator number 1. Single row of counterrotating generators located at 10 percent chord.



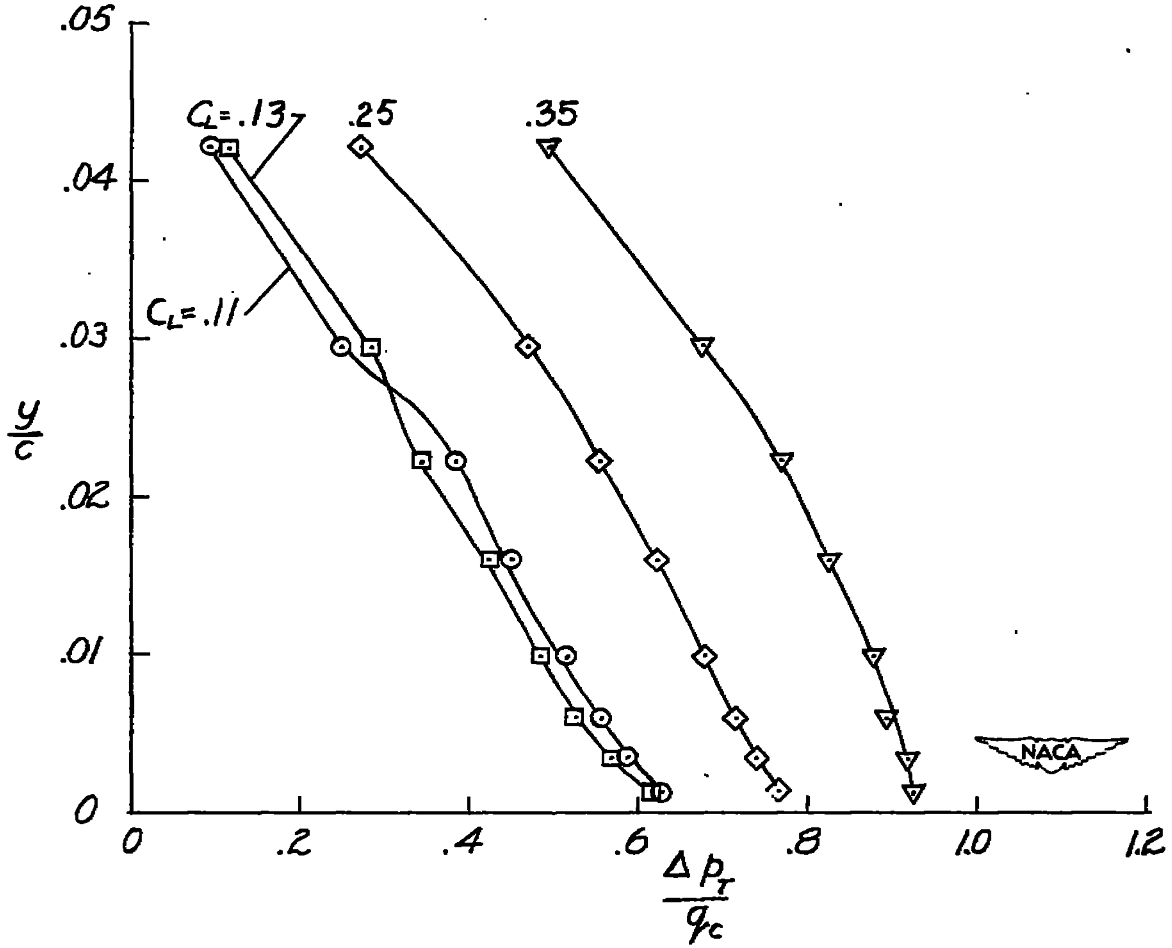
(b) $M_0 = 0.745$.

Figure 12.- Concluded.



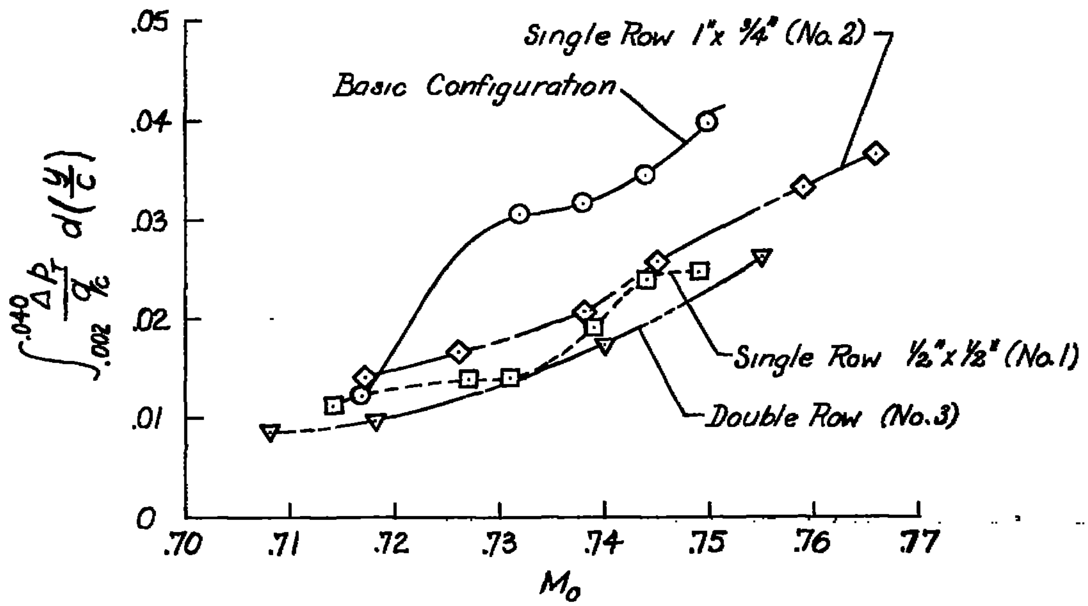
(a) $C_L = 0.17$.

Figure 13.- Variation through part of the wake at the trailing edge of the ratio of total-head loss to free-stream impact pressure. Vortex generator number 5. Single row of counterrotating vortex generators located at 30 percent chord.

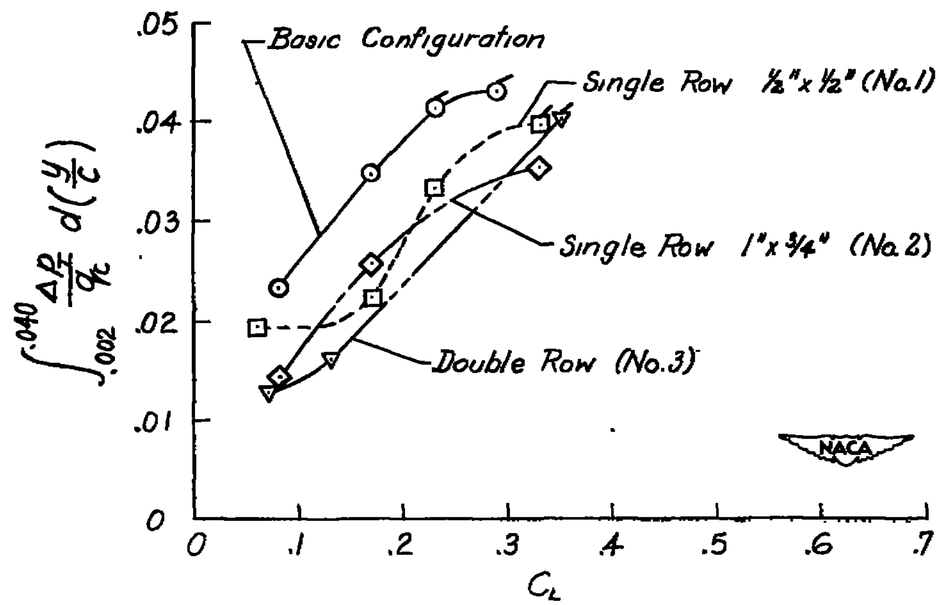


(b) $M_0 = 0.745$.

Figure 13.- Concluded.

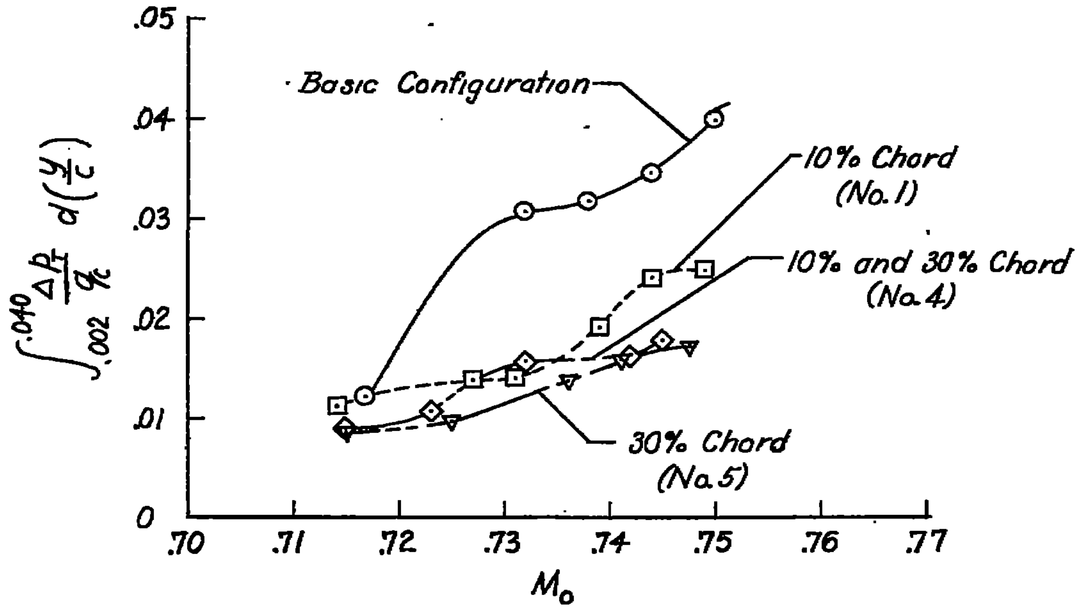


(a) $C_L = 0.17$.

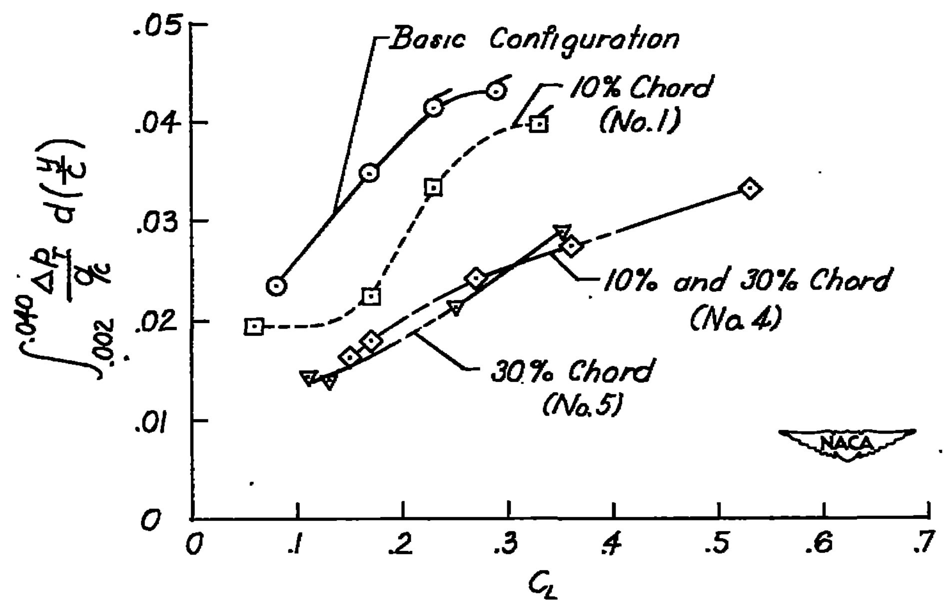


(b) $M_0 = 0.745$.

Figure 14.- Comparisons of the effects of various configurations of counterrotating vortex generators located at 10 percent chord on the integrated total-head loss at the trailing edge. Tailed points indicate flow was separated at the trailing edge.



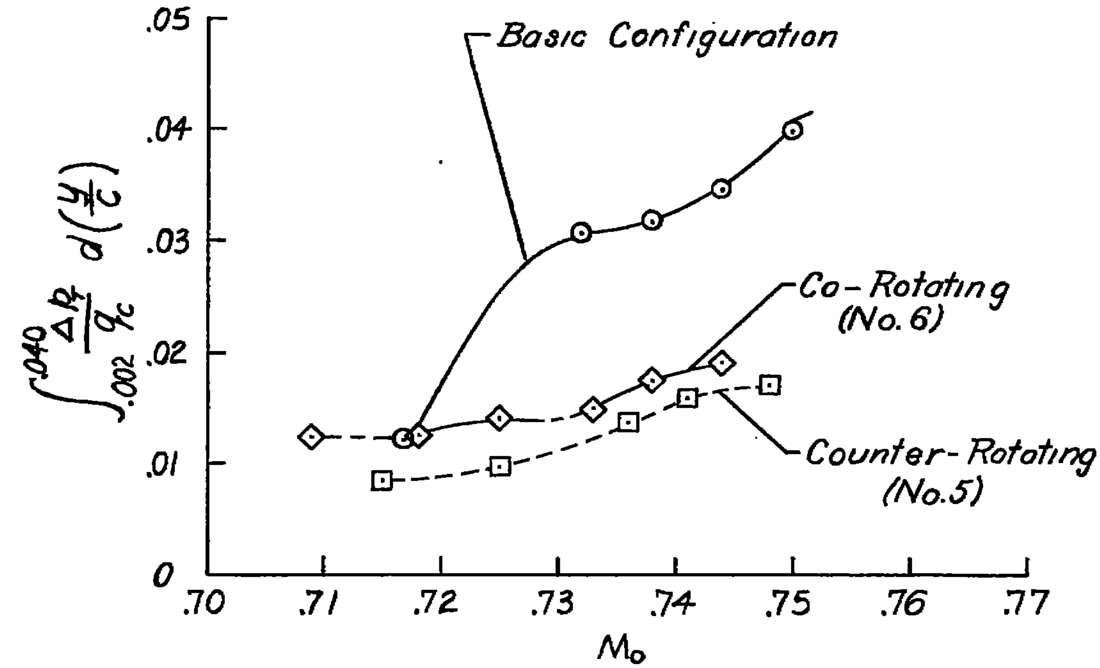
(a) $C_L = 0.17$.



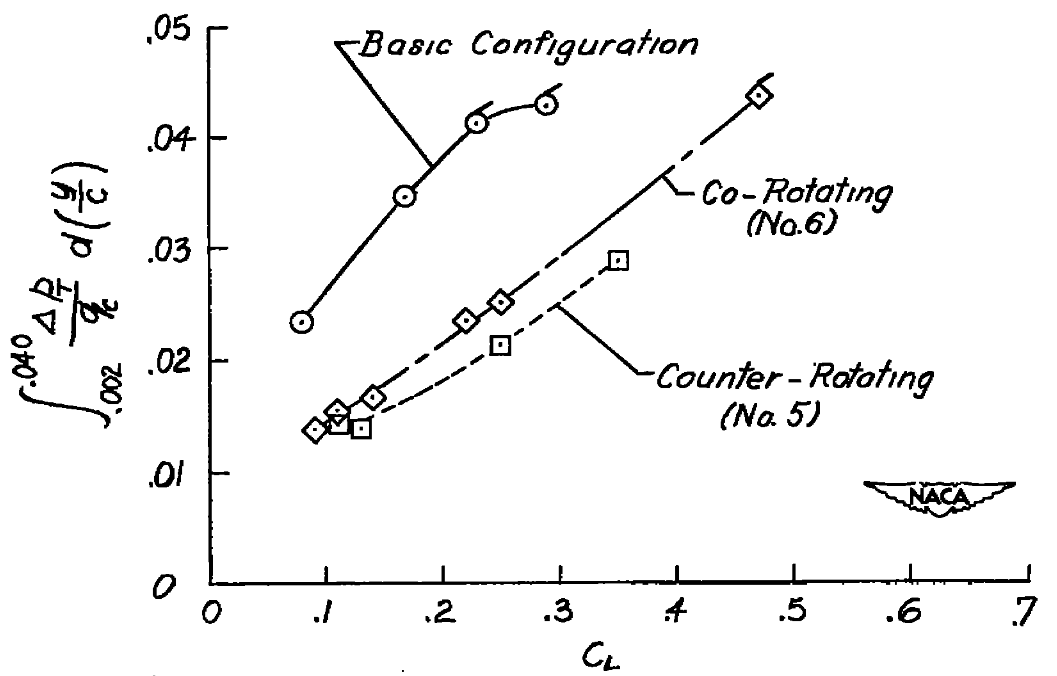
(b) $M_O = 0.745$.

Figure 15.- Comparisons of the effects of chordwise location of counter-rotating vortex generators on the integrated total-head loss at the trailing edge. Tailed points indicate flow was separated at the trailing edge.

[redacted]



(a) $C_L = 0.17$.



(b) $M_O = 0.745$.

Figure 16.- Comparison of counterrotating and co-rotating vortex generators located at 30 percent chord on the integrated total-head loss at the trailing edge. Tailed test points indicate flow was separated at the trailing edge.